SSC01-V-6

Earth Observing-1 Spacecraft Bus

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Abstract:

The Earth Observing 1 (EO-1) spacecraft is one of the most trouble-free spacecraft that NASA has launched in the last ten years. A NASA New Millennium Program (NMP) mission dedicated to validating revolutionary technologies that will be used in future government and commercial missions, EO-1 was launched in November 2000 and flies in formation with Landsat 7. As the prime contractor, Swales Aerospace designed and built the spacecraft bus, integrated and tested the EO-1 observatory, and performed launch-site operations. Developed under the faster-better-cheaper philosophy, EO-1 is an example of a successful low-cost mission. We describe the mission, its new technologies, the results of on-orbit evaluation and the keys to EO-1 success, from early design to final testing.

Overview of Earth Observing 1

This paper first describes EO-1 and then discusses its design and development. The last and largest section then provides details of the EO-1 spacecraft on-orbit performance.

Additional information on EO-1 is available in the references and at the EO-1 Internet web site, eo1.gsfc.nasa.gov

New Millennium Program

The goal of NASA's NMP is to enable 21st-century missions through identification, development, and flight validation of key breakthrough technologies so that future spacecraft can take advantage of them without assuming the risks inherent in their first use. The NMP technology development and validation process also provides a significant return of valuable science data, so that immediate benefits of NMP flights are realized along with the steady stream of new technologies for future science missions. NASA's Jet Propulsion Laboratory manages the NMP, and Goddard Space Flight Center manages the Earth Orbiter series of NMP missions. EO-1 is the first in this series of smaller, faster, cheaper Earth observing spacecraft[1, 2].

The Advanced Land Imager (ALI) instrument, built by a team under the leadership of MIT/Lincoln Laboratory, is the primary payload. The ALI is a reflective triplet telescope with multispectral detectors designed to gather the same visible and near-IR data as Landsat but with higher a signal-to-noise ratio, better spatial performance, and less cost. Details of the excellent on-orbit performance of this instrument can be found in References 3 and 4.

Mission Parameters

The Swales Aerospace EO-1 spacecraft was launched from Vandenberg Air Force Base on November 21, 2000 into a circular, sun-synchronous polar orbit at an altitude of 705 kilometers. EO-1 was co-manifested on the Boeing Delta II (7920-10C) launch vehicle with the SAC-C satellite developed by Argentina. The EO-1 orbital inclination (98.2 degrees) and descending nodal crossing time (10:01 am) puts it in "formation flight" with Landsat-7 and EOS AM-1. With all three satellites following the same ground track, EO-1 flies "behind" Landsat-7 but "ahead of" EOS-AM (Terra).

Spacecraft Capabilities

Earth Observing-1 (EO-1) Specifications

Average Orbit Power	350 W
S/C Bus Dry mass	410 kg (including WARP & X-Band PAA)
Total Mass	588 kg
Size	1.4 x 1.4 x 2m high
Payload Attach Fitting	3712
Pointing Knowledge	36 arcsec, each axis (3 sigma)
Pointing Accuracy	50 arcsec, each axis (3 sigma)
Pointing Stability (Jitter)	0.3 arcsec/sec
Slewrate	15 deg/min
ACS	Zero Momentum, 3 axis stabilized
GPS	1 receiver
Navigation Accuracy	60m, each direction (3 sigma)
Science Data Downlink capacity	105 Mb/s
Science Data Storage capability	48 Gbits win WARP
C&DH Bus Architecture	Mongoose V, Rad Hard at 12 Mhz RISC Architecture
Downlink Formats/Network	CCSDS / STDN, DSN, TDRSS
Downlink Band	S-Band (variable to 2 Mbps) X-Band (105 Mbps)
Uplink Band	S-Band (2 Kbps)
Batteries	Super NiCd / 50 Ah
Arrays	3 Panel / Si w/GaAs / Articulating / 5.25m
Nominal Voltage	28 V
Structure	Hexagonal; aluminum honeycomb
Propulsion	1 tank / 4 thrusters
Propellant Capacity	23 kg
Max delta V	85 m/s
Mission Design Life	1.5 years

EO-1 Revolutionary Technologies

Table 1 shows the ten NMP technologies that flew on EO-1. As described in a later section, two other technologies did not develop in time to meet the launch schedule and those technologies were not flown. The flight validation has been successful for most of technologies that flew on EO-1, with important flight data acquired for the remaining technologies. Particularly successful are the ALI technologies, Hyperion, and the X-band phased array. More information on the technologies can be found at the EO-1 Internet site, eo1.gsfc.nasa.gov, including contact points for detailed information.

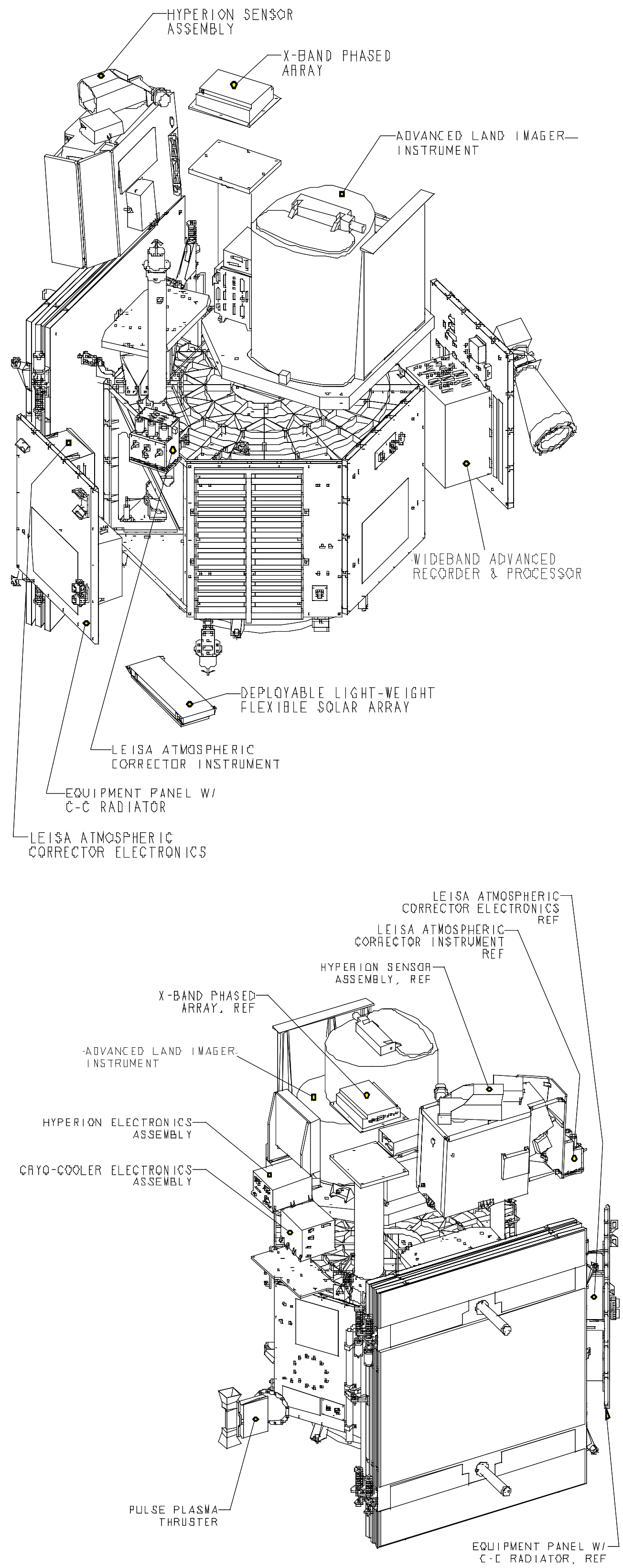

Figure 1 – EO-1 Spacecraft

To aid in risk management, NMP technologies are broken into three categories:

- Category 1 is a defining technology for the mission and is necessary for mission success. On EO-1, the ALI technologies are category 1.
- Category 2 technologies replace an existing subsystem or component that was performing a critical function. These technologies require a backup or alternate approach available in the event that the NMP technology development is unsuccessful or fails on orbit.
- Category 3 technologies receive a flight demonstration without interfering with normal operations or higher-category technologies. Category 3 technologies are often secondary payloads.

Table 1 – New Millennium Technologies on EO-1

Technology	NMP Cat.	Description
SiC optics	1	The optics are made of SiC, a high thermal conductivity, low CTE material to maintain alignment in a wide array of thermal conditions
Wide FOV, high-resolution, reflective optics	1	Telecentric optics are compact, high-resolution, and contain no moving parts; ideal for push-broom instrumentation.
Non-cryogenic detectors	1	Detectors at approximately 220 K use Thermal Electronic Control (TEC) and passive radiators rather than cryogens or mechanical pumps.
X-band phased array antenna	2	Uses an array of 64 radiating elements to focus and electronically point without gimbals. Replaces an earth-coverage antenna.
Carbon-carbon thermal radiators	2	Uses high-conductance composite materials as structural elements (composite facesheets).
Linear Etalon Imaging Spectral Array/Atmospheric Corrector	3	Measures water vapor and aerosols to correct ground images for absorbence by the atmosphere.
Pulsed plasma thrusters as attitude-control actuators	3	Low cost, low mass, high Isp propulsion system to demonstrate attitude control. ACS commands go to PPTs instead of a reaction wheel to demonstrate feasibility.
Lightweight Flexible Solar Array	3	Small secondary payload to test copper indium diselinide/CulnSe2 (CIS) solar cells & ultra-thin mylar substrate & shaped-alloy hinges & release mechanisms.
Formation flying	3	Maintain orbit with high precision relative to another satellite. Ideally, performed autonomously without ground support. Enables coordinated, stereo, & near-simultaneous imaging.
Hyperion	3	Hyperspectral imaging spectrometer, covering 400-2,500 nm with 10nm resolution.

EO-1 Development

Keys to Success

As prime contractor, Swales Aerospace designed and built the spacecraft bus, performed mission integration and test, and conducted payload integration and launch operations. Since EO-1 is a technology-demonstration mission, and since EO-1 was formulated under the faster-better-cheaper philosophy advocated by NASA during the middle 1990's, Swales Aerospace management had to accept a higher level of risk in order to reduce mission costs.

Even though the Earth Observing 1 (EO-1) mission was developed as a relatively high-risk mission, the spacecraft bus has been one of the most trouble-free busses that NASA has launched in the last ten years. With exceptional images, the instruments also have successfully demonstrated their technologies. Many people and strategies contributed to EO-1's success. Some of the most important are listed below.

- Keep the same team from feasibility studies through launch. Some the benefits include minimal re-training costs, personal ownership in the design and mission success, and reduced documentation (see the section on Problems below).
- Find dedicated and capable engineers, particularly for the subsystem leads. There is no substitute for personal responsibility in terms of efficient, high-quality work.
- Hold frequent in-house system-level reviews to identify interface and system issues early.
- Have technical expertise at the management level. Constantly monitoring progress, actively searching for problems, and developing solutions before problems become urgent.
- Prevent costly delays by relying on rapid decisions and good engineering judgment. Most design and development problems have more than one

solution. It is usually more efficient to chose a solution and make it work rather than spend time and money searching for the optimal solution.

- Perform extensive system testing with the satellite fully integrated. This should include flight scenarios and contingency testing. On EO-1, there were five comprehensive performance tests, more than five two-day simulations that included contingencies, and a second thermal-vacuum test.
- Work with operations team throughout development, integration, and test. The operations team brings expertise and external insight. See section on operations, below.
- Have thorough but flexible quality assurance based on value-added. Thoroughness helps prevent and catch errors; flexibility prevents delays from technicalities. It takes a skilled quality engineer to know which requirements or tests can be delayed or waived without adding performance risk to the program. All quality decisions were made by personnel with the engineering knowledge to assess the value added.
- Use off-the-shelf designs. In some cases, this means accepting components that do not meet all the specifications. An example is the EO-1 propulsion tank, which had been qualified with a slightly different vibration profile. Since analysis showed that the tank would survive the EO-1 environment, we waived a re-test to EO-1 specifications.

Government/Industry Teaming

During early design, the government and industry were full partners in developing the mission concept and parameters. Litton Amecom was subcontracted to Swales Aerospace, but the relationship was more of a partnership: sharing personnel, resources, and facilities. Litton also partnered with GSFC under a Space Act agreement where Litton and GSFC both invested in hardware that was flown on EO-1, with Litton marketing the hardware for other missions.

Although the government/industry relationship later evolved to a more standard relationship with GSFC managing Swales as prime contractor, the teaming arrangement continued to facilitate rapid and successful development of EO-1. Examples are:

- Swales Aerospace was committed to the same programmatic goals as GSFC: maintain the launch schedule and reduce costs where possible. This commitment was apparent in work such as the DC-DC converter work described below, and made coordinated work more effective because of the trust between the institutions.
- GSFC supplied additional engineers during thermal vacuum testing. This permitted 24-hour per day testing, which would not have been possible without the augmentation of GSFC engineers.
- Swales developed a procedure to eliminate a mechanical failure mode in the DC-DC converters used throughout EO-1 electronics. Once the issue was identified, Swales quickly developed the procedure and then applied it to every DC-DC converter on the satellite, spacecraft subsystem, payload, or GFE. This was done with a minimal impact on the schedule.
- As part of the Space Act Agreement, Litton developed the C&DH and PSE electronics and software. Sharing development costs of these items with another GSFC mission reduced EO-1 mission costs.
- When an EO-1 problem was identified, the Swales engineers worked to solve the problem as part of a combined team rather than put the entire burden on the payload provider. This was the approach, even if the problem was internal to a payload delivered to Swales. In many cases such as payload software problems, this approach saved time by developing a work-around on the spacecraft or operational procedure.
- Tecstar teamed with Swales Aerospace to develop the solar arrays. Tecstar participated in the system design and test, acquiring additional expertise in those areas, in exchange for discounting the cost of the solar cells. When EO-1 required additional power to support the addition of Hyperion, Tecstar provided, at a discount, some previously un-flown multi-junction solar cells, in exchange for flight validation of those cells.
- When the EO-1 S-band transponder failed late in environmental testing, it was too late to repair without delaying launch. GSFC provided a replacement transponder by exchanging the transponder with one intended for another GSFC mission. The EO-1 transponder was then repaired and flown on the other mission.

Hyperion Addition

Hyperion, a hyper-spectral imager built by TRW, was added to EO-1 after the failure of the LEWIS mission, which carried an instrument nearly identical to Hyperion. Hyperion restored hyper-spectral capability to EO-1, which recently had been lost when the Wedge Imaging Spectrometer was de-scoped from the ALI.

The Hyperion is a high-resolution hyper-spectral imager capable of resolving 220 spectral bands (from 0.4 to 2.5 microns) with 30-meter spatial resolution and

10-micron spectral resolution. The instrument images a 7.5-km wide swath and provides detailed spectral mapping across all 220 channels with high radiometric accuracy.

The addition of this second major instrument to the EO-1 mission late in development demonstrates that faster and cheaper also can be better. Swales Aerospace had already completed design and development of the spacecraft bus, but was able to accommodate Hyperion by quickly identifying the interfaces and issues, and then quickly finding acceptable solutions. The spacecraft was then de-integrated, with electrical and mechanical modifications completed in less than two months, and a total launch delay of six months. The changes required by the addition of Hyperion include:

- Adding another string of solar-array cells to an unpopulated portion of the solar array. Using multi-junction cells provided some margin.
- Adding a louver to the battery panel to conserve heater power that would have been necessary to keep the battery within temperature limits.
- Building platforms on the nadir deck to support the Hyperion optics assembly and two electronics boxes.
- Adding another 1773 remote terminal and associated software to communicate with Hyperion.
- Combining some of the existing power services to free-up services for Hyperion.
- Re-analyzing load, stresses, ACS, and power performance to validate the new design.
- Modifying test procedures, operational procedures, GSE, and the C&T database.

Operations Support

Two cost-saving aspects of the operations concept also reduced risk and contributed to mission success. The first was using the same GSE hardware, software, and test protocol (the ASIST system) for I&T and for on-orbit operations. The second was using the operations team to augment the spacecraft team during integration and test. Taken together, these helped build a single EO-1 team making testing and operation planning more efficient.

When the operations system is different from the I&T system, it takes personnel, funds, and time to train the flight operations team (FOT) and to convert the spacecraft C&T database to the operations format. These resources were particularly precious in EO-1's low-cost environment. Since the FOT was involved in EO-1 testing starting at box-level testing, extra training was minimal, avoiding a drain on critical I&T resources. Converting the spacecraft C&T database was simply transferring configuration control of the information, not man-years of effort to change the software and verify it.

There were numerous other benefits to this approach:

- The flight operations team (FOT) developed many of the I&T test procedures.
- The FOT was able to augment the I&T when necessary to operate additional shifts. This was particularly useful during 24-hour-per-day thermal vacuum testing.
- The FOT prepared some of the mission documentation, including writing sections of the Spacecraft Users Guide.
- During I&T—and even earlier, during design—the FOT provided additional insight and perspective that can only be provided with a detailed understanding acquired over an extended time.
- The FOT relied heavily on I&T procedures when developing flight operational procedures. The FOT benefited directly from the successes and problems that occurred during I&T.
- Since there was a role for the operations team early in the mission, the FOT was able to hire people early. Members of the FOT were able to participate earlier in the mission, and they were more familiar with EO-1 at launch than they would otherwise have been.
- New operation procedures were developed quickly because the entire EO-1 team was familiar with the command and telemetry system and able to critically review new work.

Mission Risk and De-scopes

Although risk was acceptable on EO-1 technology-development payloads, Swales Aerospace strove to limit risk on the spacecraft bus. Driven to a single-string design by cost constraints, Swales built redundancy into essential areas where this risk mitigation was warranted. Examples are mechanisms (solar-array release), the power harness, portions of the C&DH processor (two separate PROMS each contained the full flight code), and an ACS safe-hold mode installed on a separate processor.

Another risk-reduction action was extensive testing at the mission level, with all payloads and systems operating as similar as possible to their flight configurations.

In NMP technology-development missions, the area of greatest risk is before mission integration. Since the technologies are not readily available, they require development, which may not proceed as planned or expected. Even with contingency included in the development schedules, some of the technologies may not be available when required.

On EO-1, two technologies did not meet their development schedules and were removed from the mission. These were the Wedge Imaging Spectrometer (WIS) and the fiber-optic data bus (FODB). After missing milestones, the project team evaluated their progress and likely development rate. When it was clear that the technology would not be available in time for integration into the mission, the technologies were removed from EO-1. The process worked well, with critical milestones defined early and enforced when necessary.

With new technologies, there can be unexpected complication during integration. Swales Aerospace engineers spent extra time working with each technology developer to identify potential problems in interfaces or integration.

Problems During Development

Some of the problems that occurred during development of EO-1, along with suggestions on how to avoid the problems in future missions, are listed below.

- The avionics boxes were delivered late. There was evidence early in their development schedules that indicated that they would be late. Even if early delays in schedule can be accommodated by using schedule contingency, the early delays must be critically assessed to identify any larger, more pervasive problems.
- EO-1 lost key personnel, including some subsystem leads. Even on a low-cost program, each subsystem should have at least two senior or mid-level engineers to guard against personnel changes. The problem is exacerbated if documentation has been limited to reduce cost.
- EO-1 had limited documentation, an intentional decision designed to reduce costs. Although this was effective, EO-1 needed more documentation—or needed it earlier—than in the original plan. Specifically, there were many engineering decisions that were not documented. This caused the team to frequently revisit development decisions, making extra work and sometimes causing delays. Another example is the C&T handbook, which would have been helpful at the start of I&T but was not available until launch.
- In 1999, NASA began to be more risk-adverse than in the previous five years. Consequently, EO-1 had several additional reviews, received additional oversight, additional tests, and required additional documentation. This change was not anticipated, but the good engineering practices used in EO-1 development meant that there were few changes required, the most notable exception being the addition of fuses to non-critical services. If this change in philosophy had been foreseen, the primary change would have been additional documentation.
- A “cost-savings” measure that actually increased cost and schedule was eliminating Engineering Test Units for the main C&DH and power electronics (PSE). Even though every engineer and manager knows that ETUs save time, schedule, and resources, it is frequently forgotten during the heady cost-cutting days early in the program. On EO-1, the lack of ETUs caused later problems with the asynchronous timing pulse from the GPS. Lack of a power system ETU meant that the delayed power software had to be tested on the spacecraft with flight hardware.
- New development projects have uncertain schedules and require additional schedule contingency. Most of the non-NMP items on EO-1 that required development were delivered late. This includes the ACDS, PSE, software, WARP, and 1773 transceivers.
- An aggressive schedule that does not contain contingency time is only viable for well-tested components, that is, for repeated tests or repeated integration. It is convenient to manipulate schedules—and flexibility is required in near-term scheduling—but a program needs realistic and committed long-term schedule with committed interfaces to external events.

EO-1 Subsystem Description and On-Orbit Performance

Structure and Mechanisms

The EO-1 Spacecraft is a closed, hexagonal structure consisting of a top, nadir-pointing, deck incorporating the instrument payload interfaces and a bottom zenith-facing deck incorporating a transition adapter interface to the launch vehicle (see Figure 1). The zenith deck is the main interface with the hydrazine propulsion subsystem. The decks are separated by six radial supports, which transfer payload interface loads to the launch vehicle. The length of these radials supports is governed by the maximum component height with allowance for cable harness bend radii. Both the decks and radial supports are machined from AL (7075-T73). Tubular struts are used to transfer shear loads from the top to bottom deck. The transition adapter is a one piece machined AL (6061-T651) conical fitting with one flanged end which mounts to the zenith deck and the other end machined to match the Delta 3712C payload attach fitting.

The structure is closed-out along each hexagonal face by an equipment/radiator panel. Each panel is a one-inch-thick honeycomb sandwich panel. Both the facesheets and core of these panels are made from aluminum. The equipment panels are used to mount avionics components and are the heat sinks for components. Bonded aluminum edge inserts are co-cured into the panels and used to secure the panels to the structure. Post-fabrication potted inserts are cured into each panel at the desired component mounting locations. The structure is designed so that these equipment panels are easily removed during satellite I&T for access to internal subsystem components.

The structure and mechanical system of the EO-1 spacecraft are designed to meet the following requirements. The top-level requirements are to support the spacecraft subsystems and the instrument throughout the mission, with the ascent environment driving most of the design.

1. The S/C shall fit in the fairing with a minimum clearance goal of 50 mm (static envelope).
2. The S/C shall be designed to the Delta 7320 launch environment.
3. The S/C first axial & lateral mode shall be above 35 Hz and 20 Hz respectively.
4. The Spacecraft Mechanical Subsystem (SMS) shall be designed with factors of safety of 2.0 and 2.6 on yield and ultimate strength.
5. The structure shall support a maximum payload mass of 125 kg (without Hyperion), provide for its footprint and FOV.
6. Total launch mass support of 588 kg.
7. The structure will maintain the orbit alignment between the instrument mounting plate and ACS to within 0.06-degree long-term stability.
8. Provide maximize access to spacecraft subsystems during I&T.
9. The S/C first modes, in the on-orbit deployed configuration, shall be greater then 0.5 Hz for any solar array position.

Three inter-hinged panels with silicon solar cells are deployed to form a single-wing, photovoltaic solar array (S/A). The solar panels are attached to the actuator by a 3-piece tubular composite boom. Each tube is manufactured from M40J/934 composite fiber laminate.

The solar array panels are comprised of one-inch honeycomb sandwich panels with composite facesheets and aluminum core. The panel cell side is electrical insulated with a 0.002-inch layer of Kapton film. Thermal control is provided for on the non-cell side with a 0.001-inch layer of Tedlar film. Inserts are bonded along the edge of each panel to provide for GSE handling. Inserts for the solar array release system are co-cured into the panel in areas of higher density core.

The EO-1 solar array uses three nearly identical hinges at the panel-to-panel and panel-to-yoke interfaces. A unique main-deployment hinge is used at the yoke-to-actuator interface. Each hinge line uses a viscous damper to dissipate energy and a potentiometer to verify position. A constant force negator spring drives the deployment hinge, and torsion springs are used at the panel hinges. The torque margins on all hinges are at least 5 times the minimum required. The hinge designs are a derivative of the COBE, XTE and TRMM projects at GSFC. The solar array deployment is controlled by a cable and pulley system.

The EO-1 solar array is restraint during launch by a two point semi-kinematically system. One restraint assembly mounted on the S/C nadir deck and the other mounts to the zenith deck. These assemblies use High Output Paraffin (HOP) actuators. Two HOPs are used in each assembly for redundancy, only one is required to actuate for solar array deployment. Actuation of the HOPs releases a spring-loaded restraint rod that travels through the panels into a containment can located on the outer-most panel. Once the rod is released, four deck-mounted "kick-off" springs and the hinge spring torque initiate panel deployment. The restraint assemblies, including the HOP actuators, can be reset on the ground after deployment testing.

Several ground test deployments of the solar array were performed. The average HOP release time, during ground tests, was 3 minutes and the recorded on orbit time was 2 minutes and 51 seconds. Similarly, ground deployment testing of the solar array wing averaged 90 seconds as opposed to the actual on-orbit deployment of 68 seconds. On orbit release and deployment times are considered to be nominal.

Attitude Control System (ACS) Introduction

The spacecraft Attitude Control System (ACS) performs slew-and-hold maneuvers to point the body-fixed instrument for sun calibrations on an approximately weekly basis, and performs a complex series of slew-and-scan maneuvers for lunar calibrations every month.

The pointing budget for EO-1 allows a total ground targeting pointing error of 132 asec in roll, 174 asec in pitch and 122 asec in yaw. The budget includes an ACS allocation of 54 asec in roll and yaw and 108 asec in pitch for attitude determination, and 30 asec in each axis for attitude control errors. The requirement is to meet these values as a 2σ variance and the goal for imaging is to meet the same values as a 3σ variance. The mass properties growth associated with the Hyperion instrument addition was a challenge to retaining attitude control performance. The launched spacecraft mass was 571 kg and the diagonal elements of the inertia tensor were estimated as [443 179 429] kg-m^2 with the solar array deployed at the 0 degree position.

ACS Design Summary

The architecture and major components of the ACS are presented in Figure 2. All of the attitude control functions are performed within the Attitude Control and Data System (ACDS). The circuit cards of interest within the ACDS are along the center of the Figure. The primary attitude control software resides in the Mongoose 5 main spacecraft computer. The Attitude Control Electronics (ACE) provides electrical interfaces to most ACS components and hosts the Safe Hold Mode controller. Table 2 includes a description of each of the ACS components.

The fastest SA motor stepping rate is 25 Hz (0.19°/sec) and the normal daylight rate of 8 Hz can be adjusted by ±1%. SA rotation is positive/forward for "orbit day" and negative/reverse for rewinding during "eclipse".

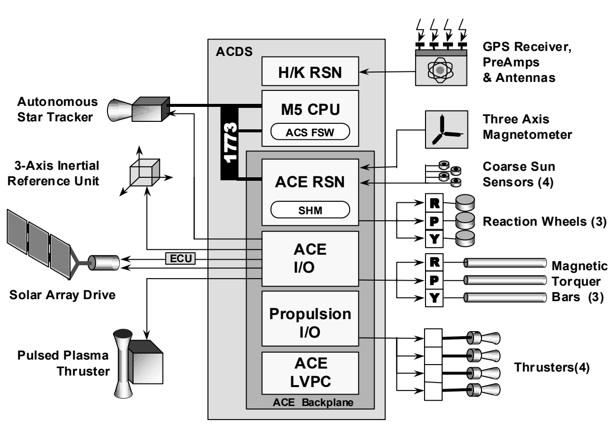

Figure 2 – EO-1 ACS Components

Table 2 – ACS Component Capabilities

ACS Component	Vendor & Model	Description
Reaction Wheels Assembly	Ithaco Space Systems Type A w/MDE	±4 N-m-s momentum at 5100 RPM, ±0.025Nm max torque; three wheels utilized with on-axis orthogonal mounting
Magnetic Torquer Bars	Ithaco Space Systems TR60CFR	±60 Am linear dipole moment with on-axis orthogonal mounting and linear coil drive capability
Three Axis Magnetometer	SAIC/Ideas and Nanotesla NT600s	±100.0 μTesla range on each of three axes; resolution of 0.05 μTesla
Inertial Reference Unit	Litton Guidance & Control SS-SIRU (with 3 HRG)	±10 degree/sec max rate; bias stability ≤0.015°/hr over 8 hours; ARW ≤0.001°/hr $^{1/2}$ (3σ); resolution of 0.05 arcsec
Autonomous Star Tracker	Lockheed Martin / ATC AST-201 with sun shade	8 x 8 deg FOV, 1 Hz ECI attitude quaternion output, with 5, 5, 25 arcsec, 1-sigma accuracy; up to 50 stars in solution
GPS Receiver	Space Systems Loral GPS Tensor with 4 antennas	1 Hz time, position to ±150 meters, 1-sigma and velocity to ±0.55 meter/second, 1-sigma
Coarse Sun Sensors	Adcole Corporation Model 29450	Peak output current of 650 micro-amps; four eyes provide full 4π steradian coverage

ACS FSW and Control Modes

The EO-1 ACS flight software architecture is derived from the GSFC Tropical Rainfall Measurement Mission (TRMM). The EO-1 ACS software includes attitude determination and closed-loop control modes for magnetic de-spin following separation from the Delta II launch vehicle, initial stabilization and sun acquisition, nadir pointed science data collection and downlink, thruster maneuvers for delta-V, and solar/lunar slew/scan maneuvers for instrument calibrations.

The ACS flight software mode transitions are illustrated in Figure 3. Following separation from the Delta II launch vehicle, the ACS nulls the tip-off rates via a B-dot magnetic control law and stabilizes the spacecraft. During initial Sun Acquisition, the spacecraft maintains an inertially fixed, solar pointing attitude with the instruments facing away from the sun. During normal operations in Mission Idle, the body fixed science instruments point toward the earth as the spacecraft maintains a fixed attitude with respect to the orbit frame. Solar calibration requires a slew maneuver to point the instruments toward the sun, followed by an Inertial Hold. A series of transitions between slew maneuvers and holds is used to perform the lunar calibration raster scan that sweeps the moon across each of the instrument detectors. A transition to Delta-V is preceded by a slew maneuver to orient the spacecraft for the thruster burn. Transition back to Earth pointing from any attitude is achieved using an Earth Acquisition slew. The ACS Fault Detection and Correction (FDC) system includes actions that can force transitions to either ACS Sun Acquisition or ACE Safe Hold.

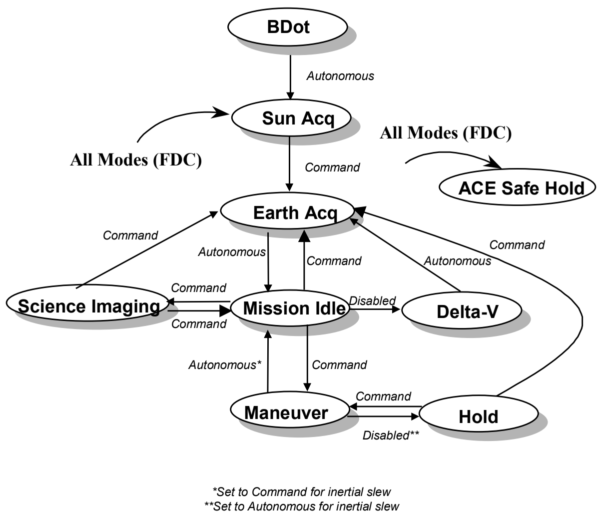

Figure 3 – ACS Control Mode Diagram

Safe Hold Mode

The independent Safe Hold algorithm software resides in the ACE RSN processor. If a fault condition is detected, this fully autonomous control capability will drive the solar array to the 0 degree reference position (orbit noon position) and put the EO-1 spacecraft in a thermal and power safe attitude that is inertially fixed relative to the sun. Coarse sun sensors on the solar array and spacecraft main body will provide spacecraft attitude with respect to the sun with the IRU providing rate feedback. The control about the sunline will only be rate damped during orbit day, but all axes will be inertially fixed during eclipse. Continuous magnetic unloading of reaction wheel momentum will be performed using a cross-product control law.

Launch Summary, Tip-off Rates and Despin

The EO-1 spacecraft separated form the launch vehicle with tip off rates of [3.61 0.42 –0.19] deg/sec. The rates decreased to [1.11 1.72 1.0] deg/sec following deployment of the solar array. The ACS B-Dot mode controller was able to decrease the spacecraft system momentum to below 4 Nms in all three axes in 78 minutes at which point the ACS transitioned to Sun Acquire mode. Sun Acquire mode was initiated at 00-326-20:42 just prior to the orbital eclipse period. The ACS controller performed angular rate damping and reaction wheel momentum dumping during the eclipse period. Upon entering orbit day, a pitch flip maneuver was performed to get the sun off of the backside of the solar array. Sun Acquisition was completed at 22:00, although Reaction Wheel momentum dumping did not complete until 23:00. The spacecraft body rates for this initial acquisition sequence are plotted in Figure 4.

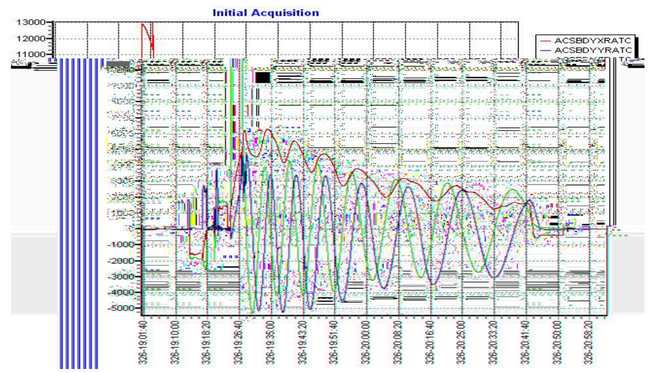

Figure 4 – EO-1 Initial Acquisition Body Rates (Arcsecond/second)

Safe Hold Mode Test

A test of the ACE independent Safe Hold Mode was planned as part of Mission Day 2 activities. The power load shed commands executed and powered off the non-essential services as expected. The ACE Safe Hold mode attitude controller exhibited excellent performance for sun pointing the solar array. The solar array sun pointing specification was 25 degrees, 3σ and the Safe Hold mode sun pointing performance, was 3 to 5 deg, 3σ including CSS albedo errors.

Earth Pointing, Mission Idle

ACS pointing performance has been well within mission requirements as shown in Table 3.

Table 3 – Science Pointing Accuracy Assessment

	Requirement	On-Orbit Performance
Controller Error	[30, 30, 30] asec, 3σ	X & Z consistently under 30asec Y varies between 0 to 50asec depending on settling time after image prep slew maneuver
Attitude Knowledge	[54, 108, 54] asec 3σ	All axes consistently under 36 asec, 3σ during normal nadir-pointed operations
Navigation Accuracy	[130m Cross-Track, 100m Along Track] 3σ	Cross-track 45m, Along track 55m, Radial 30m, 3 σ
Jitter/Rate Stability		Better than 0.5 asec/sec, 3σ during imaging

Figure 5 demonstrates that the typical attitude control accuracy achieved during an image observation meets the 30 asec, 3σ goal. The position error plot is over a 6-hour period, and the bottom plots zoom in on a 30 second image duration.

Figure 6 shows the 300 asec peak transient in the pitch axis due to opening the ALI cover prior to an image. The opening of the cover is timed such that this transient is zeroed within 2 minutes prior to the start of an image. The thermal snap transients as a result of night-to-day and day-to-night transitions have a 40 to 80 asec peak error per axis with a 2 minute width.

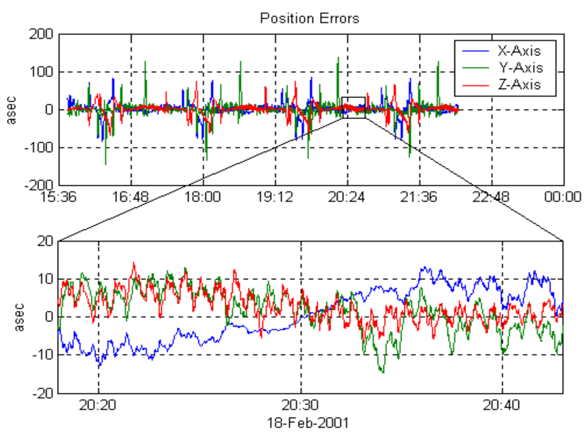

Figure 5 – Position Control Errors During Image

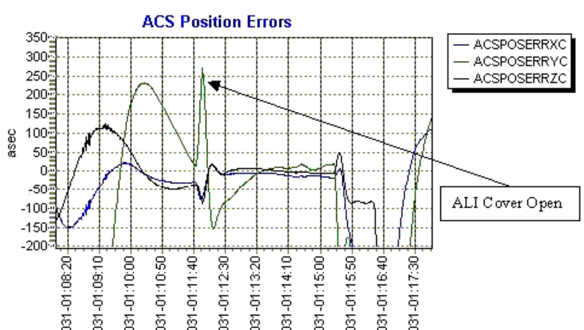

Figure 6 – Position Errors During ALI Cover Open

Delta-V Summary

Attitude control performance during thruster-based Delta-V mode has been excellent. The derived requirements for the Delta-V mode control were 5.0 degrees attitude error, and 0.2 deg/sec rate error, 3σ. The ACS phase plane plots indicate that the spacecraft is following the expected trajectories. A phase plane plot from the Day 00-346 17 minute Delta-V is shown in Figure 7. Attitude errors have predictable, steady-state values near [1.6 –1.0 0.0] deg in the X, Y, and Z-axes respectively; this is expected since there is not an integral term in the controller.

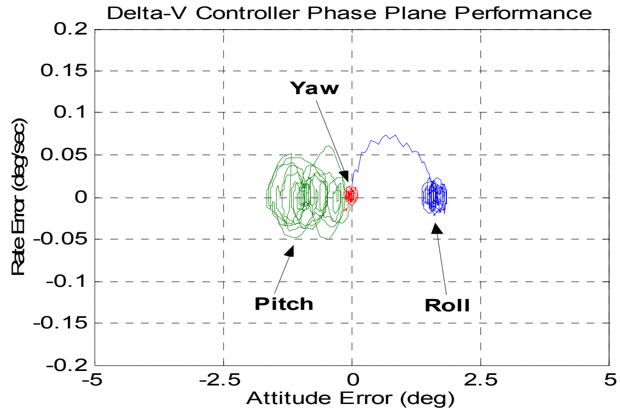

Figure 7 – Delta-V Controller Phase Plane Performance

Solar Calibration

The instrument solar calibration requires a single slew maneuver to point the instruments toward the sun followed by an inertial hold and then an earth acquisition slew back to nadir pointing.

Lunar Raster Scan

The EO-1 spacecraft is required to perform a monthly raster scan of the moon for instrument calibration. There are five detectors that must be scanned across the moon with a spacecraft pitch rate of 0.06875 deg/sec [1/8th of the normal Earth scan rate of 0.55 deg/sec]. The raster scan is made up of a series of alternating roll and pitch slews. The roll slews are used to move from one detector to another, and the pitch slews are used to scan each detector across the moon.

Lockheed Martin AST-201 Star Tracker

Overall, the EO-1 Autonomous Star Tracker performance has been better than expected. All initial AST attitude acquisitions have been successful. The mean AST RMS error has been approximately 30 urad, which correlates closely with ground processing estimates. The AST RMS error has varied between 10 and 100 asec on average with spikes as high as 500 asec. The AST has been tracking between 10 to 45 stars, with a mean of 25 stars. The AST Effective Focal Length (EFL) error telemetry indicates a mean of less than 10 µm of error with no need for updates to the EFL. The AST has shown susceptibility in the South Atlantic Anomaly, with a range of 0 to 4 events per day

– either coast frames, reacquisitions or reboots – all with autonomous recovery to track mode.

Litton G&C Space Inertial Reference Unit

The IRU has performed nominally since launch. There have been no lost packets, packet checksum errors, or invalid packets. The IRU calibration identified negligible scale factor error and the IRU-to-AST on-orbit alignment calibration resulted in less than 0.2 degree (720 asec) change. The gyro drift biases were estimated at [1.02 0.08 1.58] deg/hr and have remained constant.

Loral Space Systems GPS Tensor

The GPS receiver successfully performed two cold start acquisitions in the sun pointing orientation. The receiver remained in track during all of the IRU calibration slews, Delta-V slews, and imaging slews. It had been anticipated that the receiver may have difficulty during these off-nadir operations, but the orbit determination Kalman Filter within the GPS Tensor has demonstrated robustness through these events. The GPS position and velocity navigation vector performance has been well within the 100m spec. Reference 5 provides a more in-depth evaluation of the GPS testing and performance for EO-1.

ACS Conclusion

The EO-1 Attitude Control Subsystem design and implementation had numerous challenges as part of a 'faster, better, and cheaper' mission. The on-orbit pointing accuracy, with attitude controller errors and knowledge errors combined via a Root-Sum-Square, is approximately 40 arcsec, 3-sigma during science observations. This performance has enabled the instrument team to perform alignment calibrations using nadir scenes. The lunar raster scan maneuver has exceeded expectations and provides a significant radiometric calibration source for instrument detectors. The performance and predictability of the Delta-V mode has conserved propellant usage and set the stage for extended mission operations with the Enhanced Formation Flying experiment.

Power Subsystem Description

The EO1 Power Subsystem is a 28V, unregulated, Direct Energy Transfer (DET) power system with an articulated solar array and one 50 Ah Super nickel-cadmium (SNiCd) battery connected directly to the power bus. It regulates the spacecraft energy balance, providing the required power of 330W (nominal orbit average) while wasting very little energy inside the spacecraft to be dissipated as heat. It accomplishes this by pulse width modulating and short circuiting unwanted solar cell segments, leaving their energy out on the solar array. It distributes 5 and 15-volt regulated power internal to the Power Supply Electronics (PSE) and 28-volt unregulated power to the subsystems and instruments and provides fault protection.

Power Supply Electronics (PSE)

The PSE is contained in one electronics box consisting of the following six modules, connected to a backplane; 1 Solar Array Module (SAM), 1 Battery Module, 2 Output Modules, 1 RSN Control Module, 1 Low Voltage Power Converter (LVPC).

The PSE interfaces with the solar array, the battery and the loads. Its primary function is to provide power control (including battery charge control). But it also contains the logic and circuitry for fault protection and a telemetry and command system interface. In addition, it contains the solid state, commandable relays for power distribution of the unregulated bus power and serves as the location for the Single Point Ground.

Solar Array

The EO-1 solar array is a single wing, comprised of three panels. The array is canted at 30 degrees, and utilizes a single axis drive assembly to clock the array at an orbital rate, essentially keeping the array normal to the sun, minimizing cosine losses. A cable wrap mechanism is implemented between the rotating array and the spacecraft, which requires the array to "rewind" during eclipse periods, preparing for the next sunlit period in which it moves forward at orbital rate.

Table 4 provides details on the EO-1 solar panel design.

Table 4 – Solar Panel Summary

Item	Type	Size &/or Comment
Solar Panel Substrate	Al Honeycomb Composite	49.5 X 56.75 in.
Face Sheet	Epoxy-graphite Composite	15-mil thick
Face Sheet Insulator	Kapton	2-mil thick
Cell Bonding Adhesive	CV 2568 Silicone	NA
Thermal Control Coating	Tedlar	2-mil thick

Item	Type	Size &/or Comment
Si Cell(s)	15% BSFR Si	24.611 cm^2 3.896 X 6.317 cm
	By-pass diode protection for 31-cell sub-strings on outboard panel (diodes located on rear of panel).	
DJ GaAs (Cascade) Cell(s)	22% Effic. Dual Junction	24.312 cm^2 3.846 X 6.322 cm
	Inherent solar cell by-pass diode protection (located on the back (underside) of each cell).	
Cell Interconnect Material	Invar	Redundant
Cell Connection Method	Solder	NA
Temperature monitoring	2 Platinum resistance temperature (PRT) sensors, mounted directly opposite one another on the front and rear surface of the outboard panel.	

The combination of the two different cell technologies was a result of the addition of another payload (Hyperion) following the start of spacecraft I&T. The original solar array design consisted entirely of 15% BSFR Si 10 ohm-cm cells. The additional power requirements of the Hyperion instrument resulted in the addition of 22% DJ GaAs (Cascade) Cells on the remaining available solar panel area, resulting in an additional 1.4 amps available at the predicted operating point. These cells were wired in as a "fixed" segment, with no shunt dedicated for their control. The total Solar Array current is 22.1 Amps, BOL.

Battery

EO1 has a single 22-cell, 50 Ah, 28V Super nickel-cadmium (SNiCd) battery. [Used in conjunction with NiCd technology, 'Super' is a Trademark ™ of the Hughes Aerospace Corporation.] The voltage range was originally specified as 22 to 36 Volts (21 volts minimum at the load), consistent with the conventional NiCd technology with one battery cell failure. The actual voltage range of this battery is 24 to 34 volts (23 to 32.5 if one cell were to short, later in life).

Redundancy and Risk

EO-1 solar array cell interconnects are a triple redundant implementation, and all wiring on the solar array is dual redundant. This dual redundant wiring is carried from the solar array back to the solar array inputs at the PSE. Within the PSE, each segment has diode protection, eliminating the possibility of a short on a single string or segment from causing mission failure. Overall, the solar array implementation allows for graceful degradation and resistance to individual failures. The loss of a segment to either short or open circuit has the net effect of reducing the total power available from the solar array, allowing the spacecraft to continue operations in a potentially degraded mode.

The PSE is single string, but has features to help mitigate risk with respect to internal failures. An over-voltage protection (OVP) circuit provides protection against bus voltages over 34.5VDC by sequentially shunting array segments to maintain control. This is implemented in hardware, and has priority over the normal PSE software control loop. Also, the PSE has internal fault detection and correction that increase the likelihood of surviving system failures or upsets.

Power System Performance

Predicted performance of the EO-1 power system, as a function of solar array and battery behavior, relied on a worst-case analysis. For the solar array, the mission orbit's radiation environment was utilized to project cell degradation out to 18 months. In addition to 1MeV fluence, factors were added to the analysis to compensate for predicted coverglass darkening (UV), and other generalized losses, while adjusting the cells performance based upon thermal predictions of its operating temperature. The battery was assumed to have degraded such that voltages during the eclipse period were well below nominal values, resulting in higher inefficiencies in power conversion.

The predicted solar array current (at 35.5V) available during sunlight from launch to mission day 550 was used to determine the maximum supportable orbit average power (OAP). The OAP model included factors such as taper time, charge efficiency, battery voltage during sunlight and eclipse, and spacecraft power requirements as a function of daylight and eclipse. The limiting criteria for the maximum OAP was achieving a full energy balance within the sunlit period, i.e. achieving a 100% battery state of charge prior to entering eclipse.

Following launch and solar array deployment, EO-1 transitioned from a tumble, to sun acquisition over several orbits. During sunlit portions of the orbits, the array was able to provide some charging to the battery when the orientation of the spacecraft put the array on the sun-line, which offset the potential depth of discharge. The lowest value of battery state of charge, 85%, occurred just prior to sun acquisition. Within two orbits of sun acquisition, a full recharge of the battery occurred, providing the first energy balance on orbit. Figure 8 shows a plot of critical power system parameters from spacecraft separation from the launch vehicle, to the first orbit with an energy balance.

Over the first few weeks of the mission, the battery charging VT intercept and charge/discharge (C/D) ratio were adjusted to optimize charging performance, resulting in a VT of 4.5 (NASA standard VT curves), and a C/D of 1.05. The time duration in taper, end of taper current, and end of night battery voltage telemetry points were used to select these values, resulting in a stable end of night battery voltage, a taper time of 15 to 20 minutes (nominal), and an end of taper current of <2.5 amps. Since the first month, none of these constants have been adjusted, as the system is still behaving nominally.

Periodic events such as maneuver (thruster burns), and instrument calibrations result in a higher OAP, while at the same time result in the solar array being taken off sun pointing for a brief period of time. These occasionally result in not achieving an energy balance on the given orbit, but full balance has always been achieved within one more orbit. The nominal DCE orbit is the design and performance driver, and all analysis has been devoted to this mode.

The initial prediction for solar array current was 21.8 amps, but initial telemetry revealed a maximum array current of 24 amps. Evaluation of system performance at mission day 200 shows the power system to be still performing within mission requirements and the maximum solar array current is just over 22 amps. After adjusting the model for initial conditions, max OAP predictions can be seen in Figure 9. Radiation and UV degradation of the array have not been as severe as predicted. Figure 9 reveals that meeting the spacecraft's OAP need of 330W is achievable with significant margin through mission day 550 and beyond.

The EO-1 battery has been performing within expected tolerances, and has been demonstrating a slight downward trend on average minimum voltage (which correlates to average end of night voltage). An analysis of the data has shown a decrease in this voltage of ~590micro-volts per day. If it is assumed that this degradation is linear, it can be predicted that it will take >1400 days to reach an average minimum battery voltage of 26V, which is still within the tolerances of all spacecraft subsystems, and is well beyond the mission duration requirements.

Power System Conclusion

The power subsystem of the EO-1 spacecraft is an effective implementation of a single string design for a LEO application. The entire system continues to exceed predictions on overall performance (orbit average power support), and is capable of operations at full capacity well beyond the initial design lifetime.

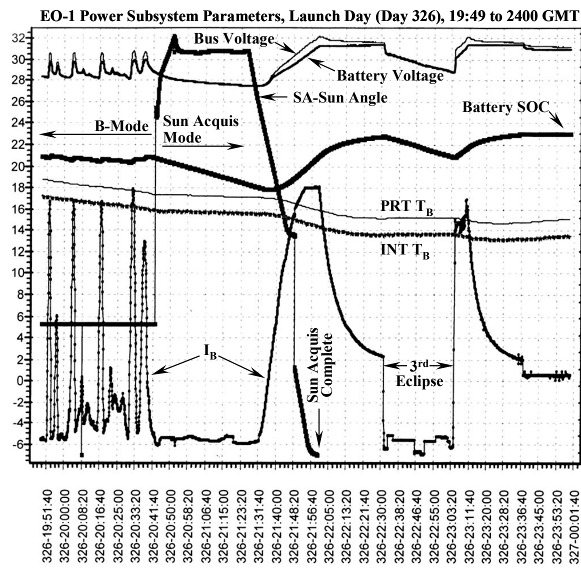

Figure 8 – Power System Parameters

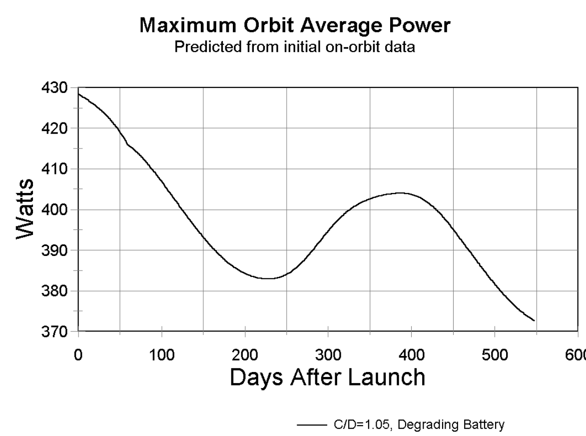

Figure 9

RF Comm

EO-1 has both S-Band and X-Band communication systems. The S-Band system provides command uplink, housekeeping data downlink, and backup science data downlink, while the X-Band system is the primary downlink for the science data. The S-Band system uses two omni antennas, one nadir and one zenith pointing, to provide near-spherical coverage. These antennas are driven by a Lockheed Martin CSX-600B transponder, which is controlled by the Comm RSN. The X-Band system uses the NMP phased-array antenna, which is steered electronically for each transmission pass over a ground station. All data formats for uplink and downlink streams conform to the CCSDS recommendations. A block diagram of the communications subsystem is provided below in Figure 10.

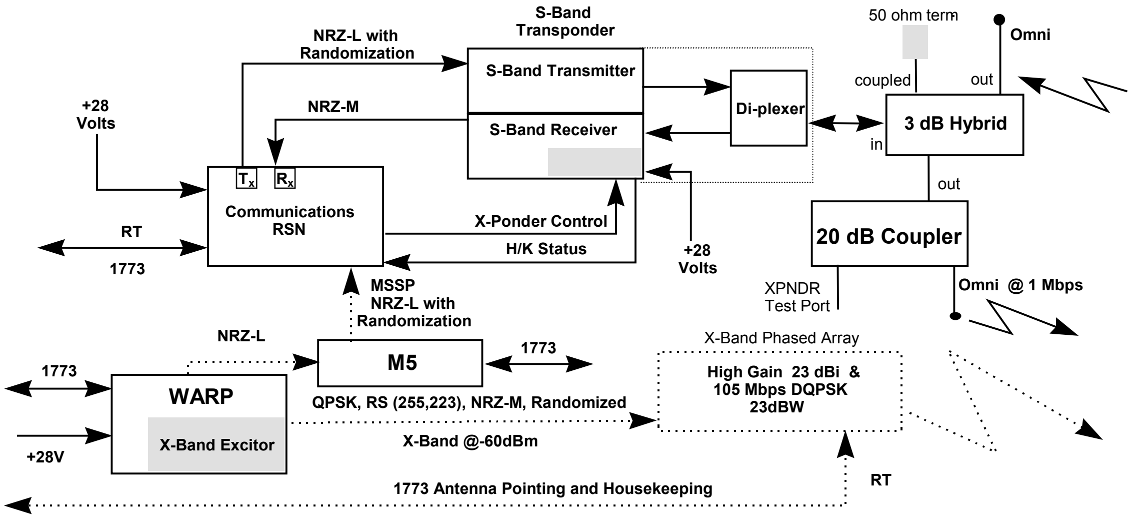

Figure 10 - Spacecraft RF Communications Systems

The primary ground station for uplink and downlink are the Spitzbergen Ground Station (SGS) located at Svalbard, Norway. During Launch and Early Orbit (L&EO) the Tracking and Data Relay Satellite System (TDRSS) was used to provide real-time housekeeping telemetry. The spacecraft requires at least two ground station passes per day, but typically uses 6 to 8 passes per day.

Command Processing

Commands originating at the EO-1 MOC are encoded into the NRZ-M format at the ground station at the rate of 2 kbps. The formatted command is used to phase shift key (PSK) modulate a 16 kHz subcarrier. A 2 kHz clock is also provided by dividing the 16 kHz clock in the ground processor by 8. The modulated subcarrier will phase modulate (PM) the uplink transmitter at the GN ground stations, which operate at a frequency of 2039.645833 MHz.

S-Band Telemetry Processing

Two independent paths transfer data from the C&DH to the Comm RSN: a Medium Speed Serial Port (MSSP, RS422) link, and the 1773 bus. Both paths feed a single physical downlink path to the transponder. The Comm RSN provides hardware based encoding of the data and determines the downlink bit rate. The MSSP link feeds high rate data from the WARP directly through hardware to the transponder. The transponder is responsible for the radio frequency transmission of the digital data stream received from the Comm RSN. Fill data is automatically generated when no data is available.

The EO-1 Spacecraft S-band transponder always operates in STDN receive mode during normal on-orbit operation. The uplink signal is received at the Spacecraft (via either omni antenna) by the S-band Transponder, which detects and phase locks to the signal swept at rates between 5 and 35 kHz/sec. The recovered command data and clock signal, along with a lock indicator, are sent to the Comm RSN for command processing and execution.

During S-band communications, the GN generates two-way Doppler (i.e., range rate) tracking data that is derived from the Spacecraft to GN S-band RF carrier, which is a coherent turnaround of the GN RF carrier. In the coherent mode, the Spacecraft's transmit carrier frequency is related to the received carrier frequency by the ratio 240/221. The GN will also generate one-way Doppler tracking data that is derived from the Spacecraft to GN S-band RF carrier. The GN provides angles data for all passes. TDRSS will generate one-way Doppler tracking data using the return link S-band RF carrier from the EO-1 Spacecraft.

Tone ranging is supported by this transponder, but this feature is not used.

S-Band Performance Testing

For the S-band portion of the system, tests include the receiver best-lock frequency, uplink acquisition threshold, uplink command threshold, downlink center frequency stability, and receiver tracking threshold.

EO-1 made extensive use of the NASA GSFC Compatibility Test Group. The compatibility testing verified that the EO-1 RF system complied with all requirements of the NASA ground network and NASA space network (TDRSS). The verification process included data flows from the observatory through a ground station simulator (via RF) and on to the EO-1 mission operations center. The verification also included RF measurements and calibrations of the EO-1 S-band system. The use of network compatibility testing was essential to the smooth operation of the RF system on orbit.

X-Band Performance Testing

The X-Band, electronically steerable phased array antenna was one of the new technology elements of the EO-1 mission. Unlike dipole-type antennas, phased array antennas are not conducive to testing with a typical hat coupler. Since, part of the verification process includes steering the antenna beam to multiple directions. The appropriate testing method for the EO-1 phased array was the near-field test technique, which provides a complete picture of the antenna pattern by performing phase measurements as well as power measurements for each array element and then performing a far-field transformation. Near-field testing was performed on the EO-1 antenna at three different periods: prior to installation of the antenna unit on the spacecraft, after re-work of the antenna data interface to the spacecraft, and after observatory thermal vacuum testing.

RF On-Orbit Performance

Overall, the EO-1 communications margins are larger than expected which is typical, since link margin calculations tend to be conservative. The S-band system operated perfectly beginning with the initial TDRSS acquisition 6.5 minutes after the EO-1 launch. The X-band flight system has performed perfectly but the ground station system had difficulty dealing with tracking the beam. At this time, all of the problems associated with the ground communication have been associated with ground systems and not the flight hardware. After the first two months of EO-1 operations, the ground station difficulties were corrected and the RF link was nominal.

RF On-Orbit Issues

After launch, two issues arose with respect to the RF system. The first issue was the tracking data products. The ground stations were using an incorrect frequency offset in the tracking data files that were sent to flight dynamics for ephemeris generation. This tracking data problem resulted in the inability of the ground stations to use program track for aligning the ground receive antennas. The second issue was the ground station not being able to autotrack the X-band signal. Later investigations found this problem resulted from a combination of incorrect ground system settings and ground equipment calibration problems. Having corrected these problems, the X-Band antenna and the ground stations are communicating nominally.

Propulsion

The EO-1 spacecraft requires an RCS for orbit adjust and precision orbit maintenance. The EO-1 RCS is a mono propellant hydrazine system operating in blow-down mode. The entire RCS is located on the lower deck of the spacecraft opposite to the payload attach fitting. The system's four MR-103G thrusters are mounted such that they fire through holes in the lower deck, and are canted at a 15-degree angle to provide coupling for spacecraft yaw control. The thrusters incorporate dual coil dual seat valves in series to mitigate the risk of valve leakage. A latch valve can be used to isolate the thrusters from the propellant tank. The RCS propellant tank is a Pressure Systems Inc. spherical titanium tank identical to the TRW STEP IV propellant tank (P/N 80225-1). This tank has a propellant capacity of 1375in^3 and an MEOP of 320 psig. An AF-E-332 elastomeric diaphragm inside the tank provides positive propellant expulsion. Nitrogen was used to pressurize the system prior to launch. The RCS is instrumented with a pressure sensor, latch valve position indicator and several temperature sensors.

A schematic of the EO-1 Reaction Control System is shown in Figure 11. The physical arrangement of propulsion system components is shown in Figure 12. Following propellant loading and pressurization, the latch valve was closed for transportation to the launch pad. The latch valve was opened prior to launch.

Temperature control for the tank, lines and isolation/latch valve, and thruster valves is done by thermostat switches that autonomously connect +28 VDC power to resistive heater loads. The thermostat switches close when cooled to some setpoint temperature and then open when warmed to some higher temperature. There are two redundant separately controlled heater system circuits. Each is powered by a separate +28V switched service from the J2 connector on the PSE LVPC. Each service is capable of providing up to about 28 W on a continuous basis.

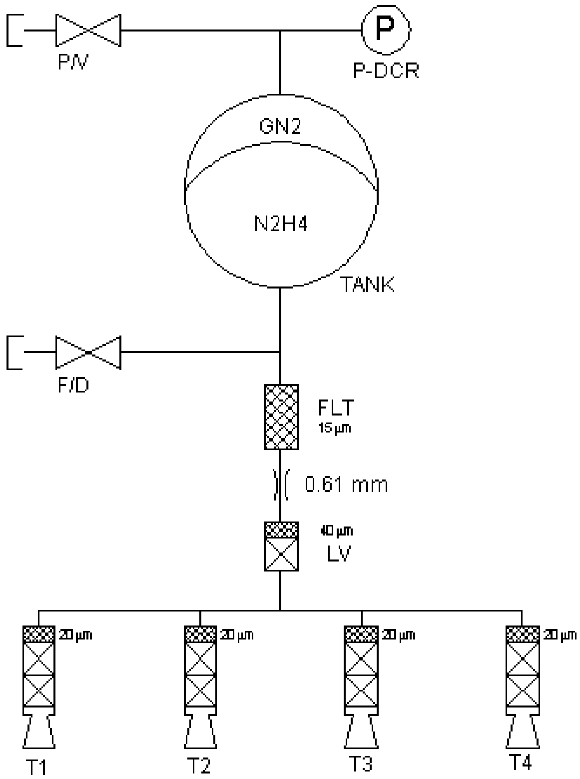

Figure 11- EO-1 RCS Schematic

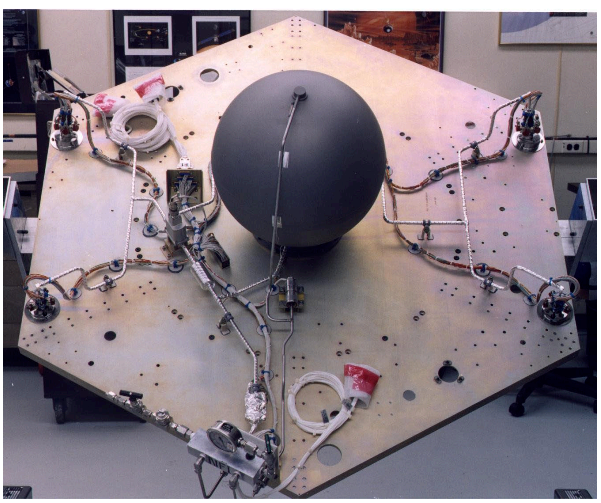

Figure 12 – EO-1 Reaction Control System

Pre-Delivery Testing

During fabrication, system components were subjected to acceptance test procedures before they were incorporated into the EO-1 propulsion system. All welds were subjected to visual and radiographic inspection. The system was verified bubble tight in accordance with EWR 127-1 paragraph 3.12.2.11.2. Prior to system acceptance, the EO-1 RCS was subjected to a General Dynamics developed acceptance test procedure (ATP). This was separate from process verification tests performed during fabrication (weld x-ray, proof testing, etc.). This ATP verified proper operation of all system components and establish a baseline for RCS parameter measurements.

Propulsion On-Orbit Performance

The orbital maneuver campaign to position the EO-1 satellite in orbit with respect to the Landsat-7 satellite was initiated on Day 4 of the mission. A 60 second calibration burn was performed to evaluate thruster operation and confirm the expected attitude control trajectories with respect to pre-launch simulations. Since the orbital separation between EO-1 and Landsat-7 continued to drift on a daily basis, the maneuver campaign required performing the first four burns relatively close to one another. The delivered delta-velocity accuracy for the calibration burn was within 5% of the prediction and all subsequent burns have been performed within 0 to 2% error. The fuel mass usage over time is shown in Figure 13.

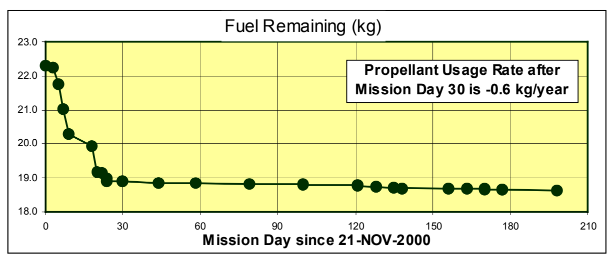

Figure 13

End of Mission Orbit Lowering

Following completion of all mission objectives, all propellant will be expended. Initial study indicates over 40 burns of 16 min duration due to the blowdown curve. Circular orbital altitude will be lowered from 705 to 590 km, with uncontrolled reentry in approximately 10 years

Command and Data Handling (C&DH)

The C&DH and attitude control system (ACS) software reside in the Attitude Control and Data System (ACDS), which contains a 12 MHz Mongoose V processor with 1.8 Gbits of storage. Most subsystems contain a Remote Services Node (RSN), which is a R000 processor for both 1773 interface and subsystem control.

Since the launch of EO-1, there have been no hardware of software anomalies associated with the C&DH

subsystem. No single event upsets have occurred, and no processor resets have been required. The only events of note are expected periodic bit errors in the C&DH random access memory, which are detected and corrected via the on-board memory EDAC. The flight ops team regularly performs table uploads into the processor memory, utilized for stored command sequences and changes to operational parameters.

The following information highlights the components of the C&DH software with respect to their on-orbit performance:

Health and Safety Monitoring

- All 15 Real-time Tasks (including ACS & EFF), in the M5 processor have performed as designed.
- No event messages or warm restarts caused by either critical or non-critical tasks failing to reporting in to health and safety on a timely basis.
- No restarts have occurred in the main onboard computer or any of the other processors on the EO-1 S/C since launch.

Telemetry Output

- All housekeeping data is being sent to the ground in real-time when in contact with the S/C as designed.
- Telemetry is consistently being routed for storage in the solid-state recorder for ground playback.
- All of the ground passes at the various telemetry downlink rates has been performed nominally.
- To date there has not been any unexplained anomalous behavior in the C&DH telemetry output string.

Time Code Management

- Maintains and distribute time with sufficient accuracy to support all aspects of the mission, including attitude determination and control and all technologies.
- Provides time keeping that has enabled precise execution of the onboard Stored command sequences.
- Provides time management for autonomously configuring S/C for ground contacts.

Command Processing

- Provides consistent and reliable commanding to support validation of all technologies and subsystems.
- Absolute and relative time stored command software performance has provided autonomous spacecraft operations.
- The stored command sequences loaded on a daily basis as an integral part of mission planning for autonomous operations.

Memory Management

- ACS has used table load capability for optimizing the attitude control parameters of the EO-1 spacecraft.
- Table loads have been used to load a new medium rate data storage filter table and to load new TSMs for failure detection and correction.
- Partial Table loads have been used to modify existing TSMs.
- The table load features are used only a daily basis to load the absolute and relative time stored command sequences for mission planning.
- To date there has been no problems identified with the memory management s/w during the on orbit checkout.
- EFF has used memory load capability to load new software algorithms.

Data Storage

- All engineering data, GPS data, and, spacecraft's events are being stored in the solid-state recorder as designed.
- The Flight Operations Team (FOT) is consistently able to playback and dump the stored data down to the ground during the scheduled ground passes.
- The command and control of the Data storage features has worked as designed without any unexplained anomalous behavior.

Anomaly Management

- All single-bit errors have been corrected when detected in the DRAM memory and no multi-bit errors have been detected.
- No Memory Checksum failures have been detected in any of the processors.
- All events that are being generated are being stored in the recorder memory and are being sent to the ground as real time telemetry when the S/C is in ground contact
- TSM task has responded to occurrence of spacecraft anomalous conditions as designed.

Enhanced Flying Formation (EFF)

- Attitude determination and control in formation with Landsat 7 has been successfully demonstrated.
- EFF flight software has been successfully activated and capabilities demonstrated.
- M5 CPU utilization peaks at over 100% CPU (as expected) utilization while EFF processes its data with no anomalous behavior onboard the spacecraft.

1773 Bus Communications Control

- 1773 Bus Control software is consistently synchronizing bus transactions for all of the unique processors in the system acting as remote terminals (RT)
- Routing telemetry and commands to and from all of the RTs has been performed flawlessly with no commands or telemetry being dropped.
- There has been no unexplained bus errors or retries in any of the RTs that are part of the EO-1 S/C.

Housekeeping (HK) RSN

- Successfully detected Launch Vehicle Separation and initiated the Solar Arrays deployment sequence precisely as designed.
- Provides control and monitoring of the Light Weight Flexible Solar Array (LFSA) as required for the technology validation.
- Collection of S/C housekeeping thermistor data that includes Structural temperatures, Carbon-Carbon Radiator temperature, and other thermistors throughout the S/C.
- Providing consistent management and control of the GPS receiver to acquire GPS State Vectors, GPS time acquisition and distribution, and data collection on request for technology validation.
- One significant issue in the HK GPS data collection software that cause the HK RSN to erroneously mark valid packets as being invalid persisting long enough for the ACS to quit using GPS for attitude control was identified. An Operational work around has been implemented in the ACS Software that negates the problem.

Comm RSN

- Telemetry downlink of science and housekeeping data from the spacecraft has been successfully performed at all programmable rates.
- The ability to downlink WARP data through the S-Band antenna has been successfully demonstrated as the downlink source for the first EO-1 images to the ground.
- No unexplained anomalous behavior has been identified in the performance of the software.

The EO-1 C&DH subsystem has performed almost flawlessly since launch. The only projected maintenance is standard flight operations tasks, and adjustment to TSM's as other subsystem performance changes with spacecraft life.

EO-1 Thermal Design

The EO-1 subsystem is designed with a cold-biased passive thermal control system (TCS). The thermal hardware includes: thermostatically controlled heaters, multi-layer insulating (MLI) blankets, thermistors, thermal louvers and optical coatings. The majority of the spacecraft exterior is covered with MLI to minimize heat loss to space and to reduce the effects of incident solar and earth flux. Thermal louvers are used on the battery equipment panel to reduce heater power and help maintain the desired battery temperature.

The majority of thermal energy is transferred from electronic boxes to the spacecraft equipment panels by conduction at the box/structure interface. Cho-Therm™ is used at the box/structure interface to enhance the heat transfer at the spacecraft interface. In addition, the boxes that contain heat dissipating components within the spacecraft have high emittance (>0.8) coatings on the outside surfaces to help maximize internal radiation heat transfer and reduce thermal gradients. The equipment panels perform as radiators by placing a specific amount of Silver Teflon on the space viewing side of the panels and radiating the excess energy to space.

For all the radiators, the radiating area is determined by using hot-case assumptions and allowing the radiator to dissipate the internal energy and absorbed environmental flux while maintaining the mounting surface below 40°C. Depending on the amount of sun exposure and the thermal-dissipation of the mounted components, between 10 and 60 percent of each panel is dedicated to radiating heat. The radiator temperature is maintained above 0°C using thermostatically controlled heaters and 5 mil silver Teflon as the radiator thermal coating. The radiator sizes were verified during the spacecraft level thermal balance test. The Battery radiator panel consists of thermal isolation using G10 spacers and includes a louver for battery temperature control because of special requirements (Table 5).

Thermal Design Requirements

The spacecraft thermal design is based on meeting the temperature requirements in Table 6.

Table 5 – Component Thermal Requirements

Description	Operational	Survival
Propulsion	+10°C to +40°C	+5°C to +50°C
Battery	+0°C to +20°C	-20°C to +35°C
Star Tracker	+13°C to +23°C	-10°C to +35°C

Some components have specific temperature control requirements. The EO-1 NiCd battery, the hydrazine propulsion system and the Autonomous Star Tracker (AST) have identified specific temperature requirements. The thermal designs for the components listed in Table 5 are tailored so they meet the thermal requirements for all phases of the mission.

Table 6 – Nominal Thermal Requirements

Description/Limits	Hot	Cold
Survival	50°C	-10°C
Qualification/Operational	40°C	5°C
Acceptance	35°C	5°C
Design/Predictions	30°C	10°C

Thermal System On-Orbit Performance

The EO-1 TCS has been nominal since launch. While on the launch pad the EO-1 battery kept cool by receiving controlled fairing air. During launch/ascent the battery TCS performed as predicted, increasing less than a 1°C at the completion of spacecraft ascent. Achieving the predicted temperatures on the HOPS actuators assisted in a successful solar array deployment. In fact, the performance of the EO-1 TCS has been excellent, and no changes have been made to the nominal TCS configuration since launch, and none are expected. The EO-1 instruments, ALI, Hyperion and LEISA/AC have all reported that the interface temperatures are as predicted and the individual thermal performance of each instrument is excellent. All NMP technologies have also reported nominal thermal conditions.

The success of the EO-1 TCS can be attributed to keeping the TCS simple: radiators, thermal coatings, heaters and louvers. Since the thermal analysis/design is only as good as the information provided by the other subsystems. Obtaining accurate power dissipations, controlling the thermal interfaces and measuring the material thermal properties are essential in achieving an accurate thermal model and thermal design. Following this philosophy the EO-1 TCS should remain stable through the end of the mission.

Conclusion

The on-orbit performance of the Swales Aerospace EO-1 spacecraft bus has been nearly flawless, meeting or exceeding all functional and performance requirements. This was accomplished despite severe programmatic constraints and the challenges of a technology-driven NMP mission.

Acknowledgements

The authors thank the Goddard Space Flight Center and all members of the EO-1 team for their roles in the mission's success. We also thank Gary Smith for his work on the C&DH and software portions of this paper, Ralph Sullivan for power, Tom Spangler for RF.

Author Biographies

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Mr. Bruce F. Zink received a B.S.E.E degree in 1986 from the University of Maryland, College Park, and M.S.E.E from The Johns Hopkins University in 1997. He has been with Swales Aerospace since 1991, where he has supported various spacecraft related activities, including responsibility for the EO-1 power subsystem, and the role of Lead Electrical Engineer on the EO-1 spacecraft. He presently manages the Electrical Systems group at Swales.

Mr. Michael J. Cully, PE, BCE Manhattan College 1979, MS Rutgers University 1984, MBA Villanova University 1989. Mr. Cully joined Swales Aerospace in 1996 as the EO-1 Program Manager. He is now the Director of Space Programs.

Mr. Michael McCullough has over 14 years of experience in aerospace systems engineering design and development of propulsion subsystems for a variety of spacecraft. He was the Lead Propulsion Engineer for several DoD and NASA programs including: Science Package 5 (SP-5) for SDI Mission Delta-181; The Advanced Composition Explorer (ACE) spacecraft for NASA; the Near Earth Asteroid Rendezvous (NEAR) spacecraft for NASA; and Earth Observing One (EO-1) spacecraft for NASA. In addition to his strong background in spacecraft propulsion he is also experienced in the development of dynamic simulators for hardware in the loop testing of spacecraft systems.

Mr. Paul Sanneman is currently the Guidance, Navigation & Control Group Manager at Swales Aerospace in Beltsville, MD. He earned his BSEE at Worcester Polytechnic Institute (WPI) in 1986 and MSEE at University of Southern California (USC) in 1989. He was the lead Attitude Control Subsystem engineer for the New Millennium EO-1 spacecraft from July 1996 through launch and checkout in Dec 2000. Other NASA science missions to which he has contributed include TOPEX/Poseidon, EOS Terra, Landsat-5, Galileo and Magellan.

Mr. Nicolas Teti has a B.S.M.E., from University of Maryland, and worked in the Thermal Engineering Branch of GSFC for 5 years before joining Swales Aerospace. He was the Lead Spacecraft Thermal Engineer on Earth Observing-1 and has worked on many other ELV and Shuttle payloads.

Mr. Peter Alea holds a BS in Engineering from Florida Institute of Technology and has completed graduate level courses in Structural Dynamics from the George Washington University. He has been with Swales Aerospace since 1987 working as a lead mechanical systems engineer on the Hubble Space Telescope repair missions and the Earth Observer 1 satellite. Before Swales, Mr. Alea worked seven years for NASA at the Goddard Space Flight Center as a test and evaluation engineer where he was responsible for instrument and spacecraft level dynamic testing on various flight programs such as OSS-1, SPARTAN, HST and COBE.

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